RESEARCH MEMORANDUM

LIFT, DRAG, AND PITCHING MOMENT OF LOW-ASPECT-RATIO WINGS

AT SUBSONIC AND SUPERSONIC SPEEDS - PLANE

TRIANGULAR WING OF ASPECT RATIO 2

WITH NACA 0005-63 SECTION

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LIFT, DRAG, AND PITCHING MOMENT OF LOW-ASPECT-RATIO WINGS AT SUBSONIC AND SUPERSONIC SPEEDS - PLANE TRIANGULAR WING OF ASPECT RATIO 2 WITH NACA 0005-63 SECTION

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SUMMARY

A wing-body combination having a plane triangular wing of aspect ratio 2 and NACA 0005-63 sections in streamwise planes has been investigated at both subsonic and supersonic Mach numbers. The lift, drag, and pitching moment of the model are presented for Mach numbers from 0.24 to 0.95 and from 1.30 to 1.70 at a Reynolds number of 3.0 million. The variations of the characteristics with Reynolds number are also shown for several Mach numbers.

INTRODUCTION

A research program is in progress at the Ames Aeronautical Laboratory to ascertain experimentally at subsonic and supersonic Mach numbers the characteristics of wings of interest in the design of high-speed fighter airplanes. Variations in plan form, twist, camber, and thickness are being investigated. This report is the second of a series pertaining to this program and presents results of tests of a wing-body combination having a plane triangular wing of aspect ratio 2 and NACA 0005-63 sections in streamwise planes. Results from the first investigation in this program are presented in reference 1. As in that reference, the data are presented herein without analysis to expedite publication.

NOTATION

b wing span, feet

$\bar{c}$	mean aerodynamic chord $\left(\dfrac{\int_0^{b/2} c^2\,dy}{\int_0^{b/2} c\,dy}\right)$, feet
c	local wing chord, feet
l	length of body including portion removed to accommodate sting, inches
$\frac{L}{D}$	lift-drag ratio
$\left(\frac{L}{D}\right)_{max}$	maximum lift-drag ratio
M	Mach number
q	free-stream dynamic pressure, pounds per square foot
R	Reynolds number based on mean aerodynamic chord
r	radius of body, inches
r_o	maximum body radius, inches
S	total wing area including the area formed by extending the leading and trailing edges to the plane of symmetry, square feet
x	longitudinal distance from nose of body, inches
y	distance perpendicular to plane of symmetry, feet
α	angle of attack of the body axis, degrees
C_D	drag coefficient $\left(\frac{\text{drag}}{qS}\right)$
C_m	pitching-moment coefficient about the 25-percent point of the wing mean aerodynamic chord $\left(\frac{\text{pitching moment}}{qS\bar{c}}\right)$
C_L	lift coefficient $\left(\frac{\text{lift}}{qS}\right)$
$\frac{dC_L}{d\alpha}$	slope of the lift curve measured at zero lift, per degree
$\frac{dC_m}{dC_L}$	slope of the pitching-moment curve measured at zero lift

APPARATUS

Wind Tunnel and Equipment

The experimental investigation was conducted in the Ames 12-foot pressure wind tunnel and in the Ames 6- by 6-foot supersonic wind tunnel. In each wind tunnel the Mach number can be varied continuously and the stagnation pressure can be regulated to maintain a given test Reynolds number. The air in these tunnels is dried to prevent formation of condensation shocks. Further information on these wind tunnels is presented in references 2 and 3.

The model was sting mounted in each tunnel, the diameter of the sting being about 85 percent of the diameter of the body base in the 12-foot wind tunnel and 73 percent of the diameter of the body base in the 6- by 6-foot wind tunnel. The pitch plane of the model support was vertical in the 12-foot wind tunnel and horizontal in the 6- by 6-foot wind tunnel. A balance mounted on the sting support and enclosed within the body of the model was used to measure the aerodynamic forces and moments on the model. The balance was the 4-inch-diameter, four-component, strain-gage balance described in reference 4.

Model

A photograph of the model mounted in the Ames 12-foot pressure wind tunnel is shown in figure 1. A plan view and front view of the model and certain model dimensions are given in figure 2. Other important geometric characteristics of the model are as follows:

Wing

Aspect ratio	2
Taper ratio	0
Airfoil section (streamwise)	NACA 0005-63
Total area, S, square feet	4.014
Mean aerodynamic chord, $\overline{c}$, feet	1.889
Dihedral, degrees	0
Camber	None
Twist, degrees	0
Incidence, degrees	0
Distance, wing-chord plane to body axis, feet	0

Body

Fineness ratio (based upon length, l, fig. 2) 12.5
Cross-section shape . . . . Circular
Maximum cross-sectional area, square feet 0.204
Ratio of maximum cross-sectional area to wing area . 0.0509

The wing was constructed by covering a steel spar with a tin-bismuth alloy. The body spar was also steel but was covered with aluminum. The surfaces of the wing and body were polished smooth.

TESTS AND PROCEDURE

Range of Test Variables

The characteristics of the model as a function of angle of attack were investigated for a range of Mach numbers from 0.24 to 0.95 in the Ames 12-foot pressure wind tunnel and from 0.60 to 0.90 and from 1.30 to 1.70 in the Ames 6- by 6-foot supersonic wind tunnel. The major portion of the data was obtained at a Reynolds number of 3.0 million. Data were also obtained for Reynolds numbers up to 15.0 million at low subsonic Mach numbers and up to 7.5 million at high subsonic and at supersonic Mach numbers.

Reduction of Data

The test data have been reduced to standard NACA coefficient form. Factors which affect the accuracy of these results and the corrections applied are discussed in the following paragraphs.

Tunnel-wall interference.- Corrections to the subsonic results for the induced effects of the tunnel walls resulting from lift on the model were made according to the methods of reference 5. The numerical value of these corrections (which were added to the uncorrected data) was, for the results obtained from the 12-foot wind tunnel:

$$\Delta\alpha = 0.265\ C_L$$

$$\Delta C_D = 0.0046\ C_L^2$$

and, for the results obtained from the 6- by 6-foot wind tunnel:

$$\Delta\alpha = 0.932\ C_L$$

$$\Delta C_D = 0.0162\ C_L^2$$

No corrections were made to the pitching-moment coefficients.

The effects at subsonic speeds of constriction of the flow by the tunnel walls were taken into account by the method of reference 6. The correction was calculated for conditions at zero angle of attack and was applied throughout the angle-of-attack range. At a Mach number of 0.95 in the 12-foot wind tunnel this correction amounted to a 2-percent increase in the Mach number over that determined from a calibration of the wind tunnel without a model in place. In the 6- by 6-foot wind tunnel at a Mach number of 0.90, the correction was somewhat larger, being 4 percent.

For the tests at supersonic speeds the reflection from the tunnel wall of the Mach wave originating at the nose of the body did not cross the model. No corrections were required, therefore, for tunnel-wall effects.

Stream variations.- Calibration of the 12-foot wind tunnel has shown that in the test region the stream inclination determined from tests of a wing spanning the tunnel, with the support system at 0° angle of attack, is less than 0.08°. The variation of static pressure is less than 0.2 percent of the dynamic pressure. No correction for the effect of these stream variations was made.

Tests at subsonic speeds in the 6- by 6-foot supersonic wind tunnel of the present symmetrical model in both the normal and the inverted positions have indicated no stream curvature or inclination in the pitch plane of the model. No measurements have been made at subsonic speeds, however, of the stream curvature in the yaw plane. At subsonic speeds, the longitudinal variation of static pressure in the region of the model is not known accurately at present, but a preliminary survey has indicated that it is less than 2 percent of the dynamic pressure. No correction for this pressure variation was made.

A survey of the air stream in the 6- by 6-foot wind tunnel at supersonic speeds (reference 3) has shown a stream curvature only in the yaw plane of the model. The effects of this curvature on the measured characteristics of the present model are not known, but are believed to be small as judged by the results of reference 7. The survey also indicated that there is a static-pressure variation in the test section of sufficient magnitude to affect the drag results. A correction was added to the measured drag coefficient, therefore, to account for the longitudinal buoyancy caused by this static-pressure variation. This

correction varied from as much as −0.0008 at a Mach number of 1.30 to +0.0009 at a Mach number of 1.70.

Support interference.– At subsonic speeds the effects of support interference on the aerodynamic characteristics of the model are not known. For the present tailless model, it is believed that such effects consisted primarily of a change in the pressure at the base of the model. In an effort to correct at least partially for this support interference the base pressure was measured and the drag data were adjusted to correspond to a base pressure equal to the static pressure of the free stream.

At supersonic speeds the interference of the sting on the body for a body-sting configuration similar to that of the present model is shown by reference 8 to be confined to a change in base pressure. The previously mentioned adjustment of the drag for base pressure, therefore, was also applied at supersonic speeds.

RESULTS

The results are presented in this report without analysis in order to expedite publication. Figure 3 shows the variation of lift coefficient with angle of attack and the variation of drag coefficient, pitching-moment coefficient, and lift-drag ratio with lift coefficient at a Reynolds number of 3.0 million and at Mach numbers from 0.24 to 1.70. The effect of Reynolds number on the aerodynamic characteristics at Mach numbers of 0.24, 0.60, 0.80, 1.30, and 1.70 is shown in figure 4. The results presented in figure 3 have been summarized in figure 5 to show several important parameters as functions of Mach number. The slope parameters in this figure have been measured at zero lift.

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National Advisory Committee for Aeronautics,
Moffett Field, Calif.

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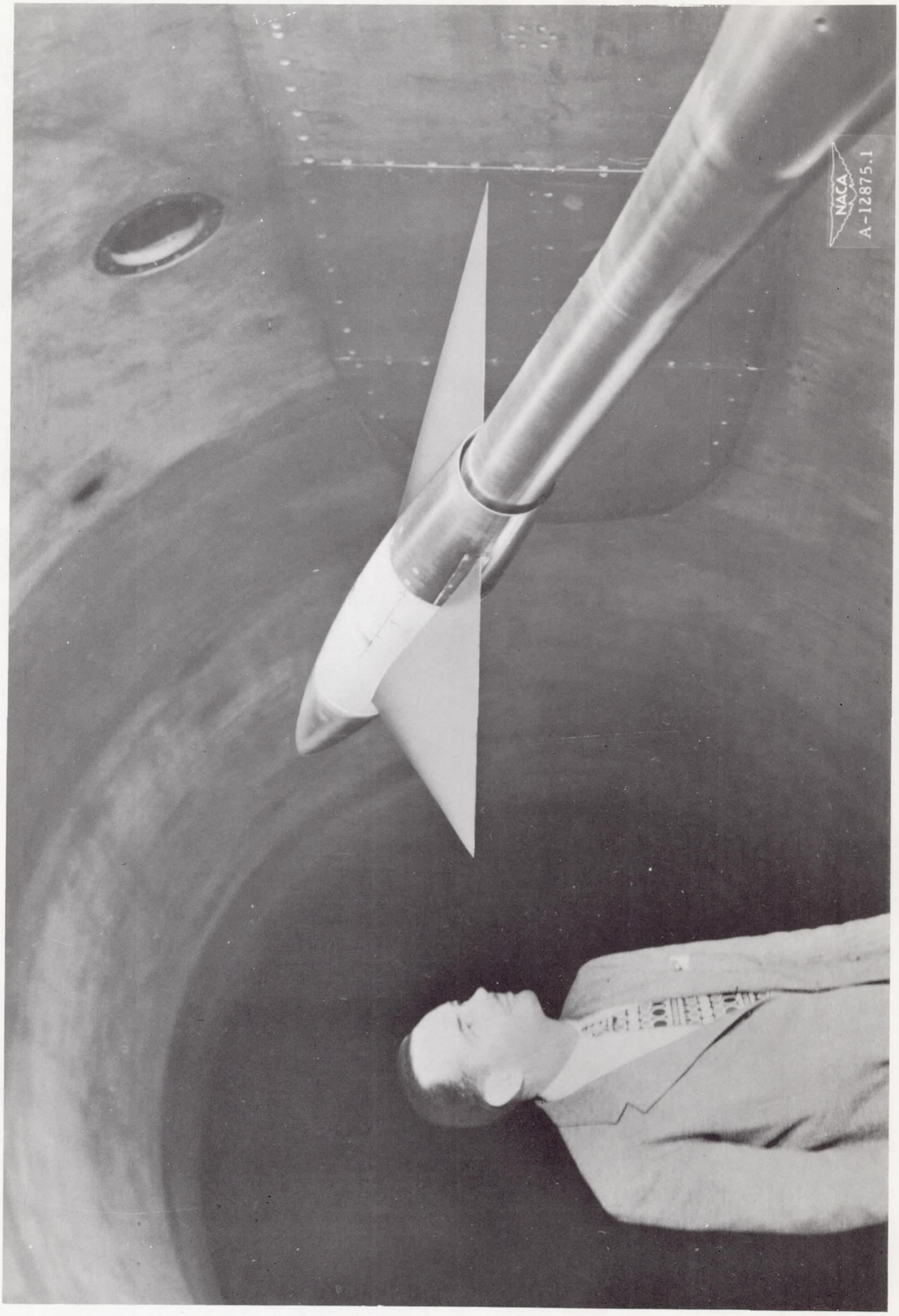


Figure 1.- The model in the Ames 12-foot pressure wind tunnel.

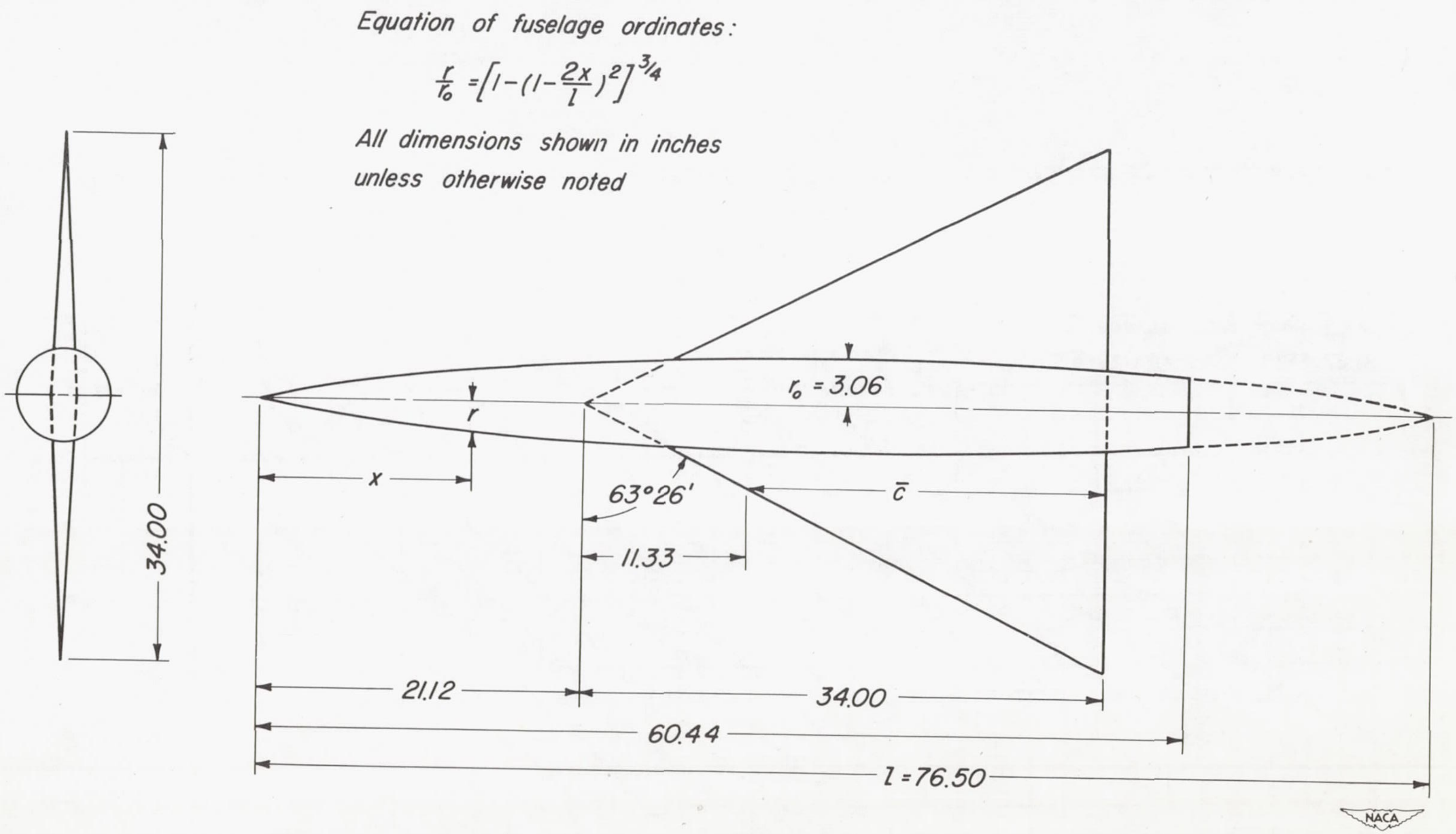
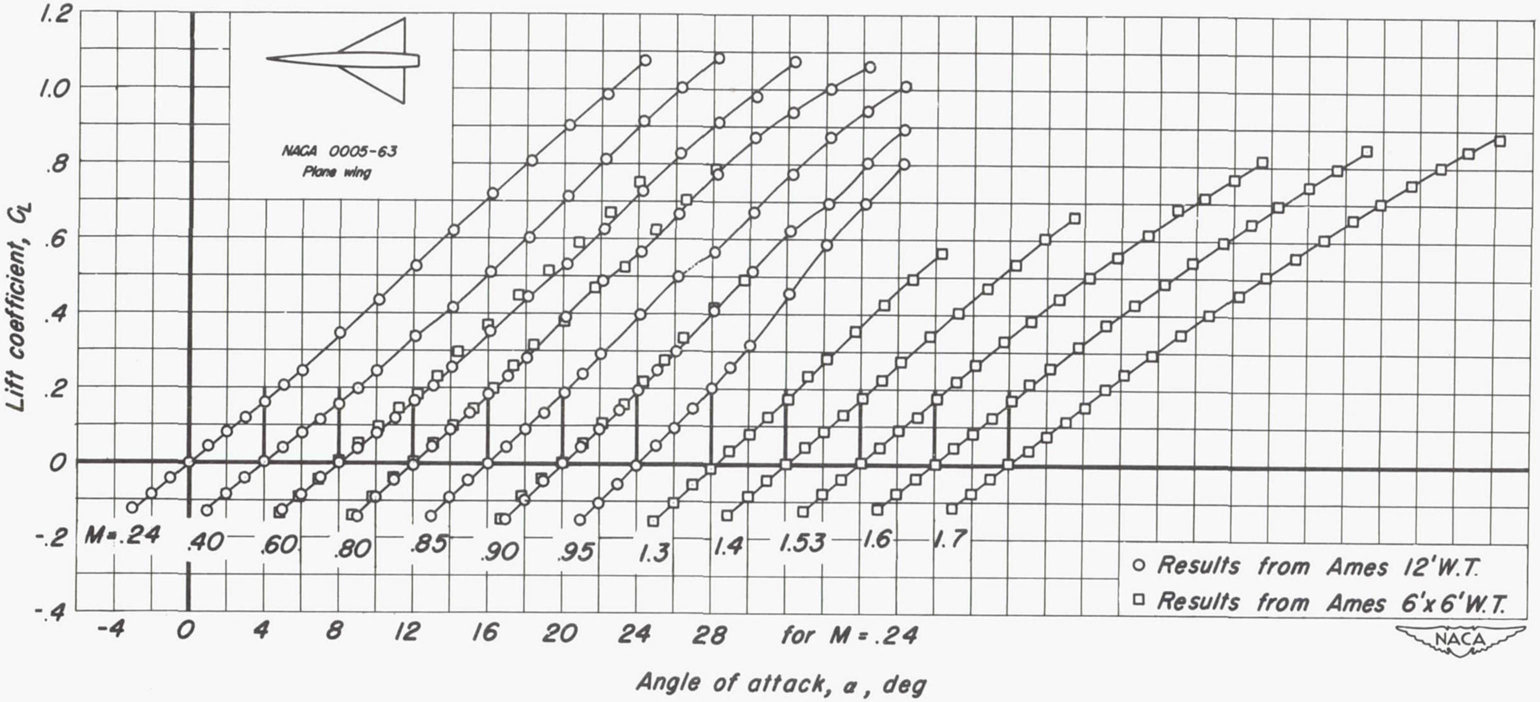
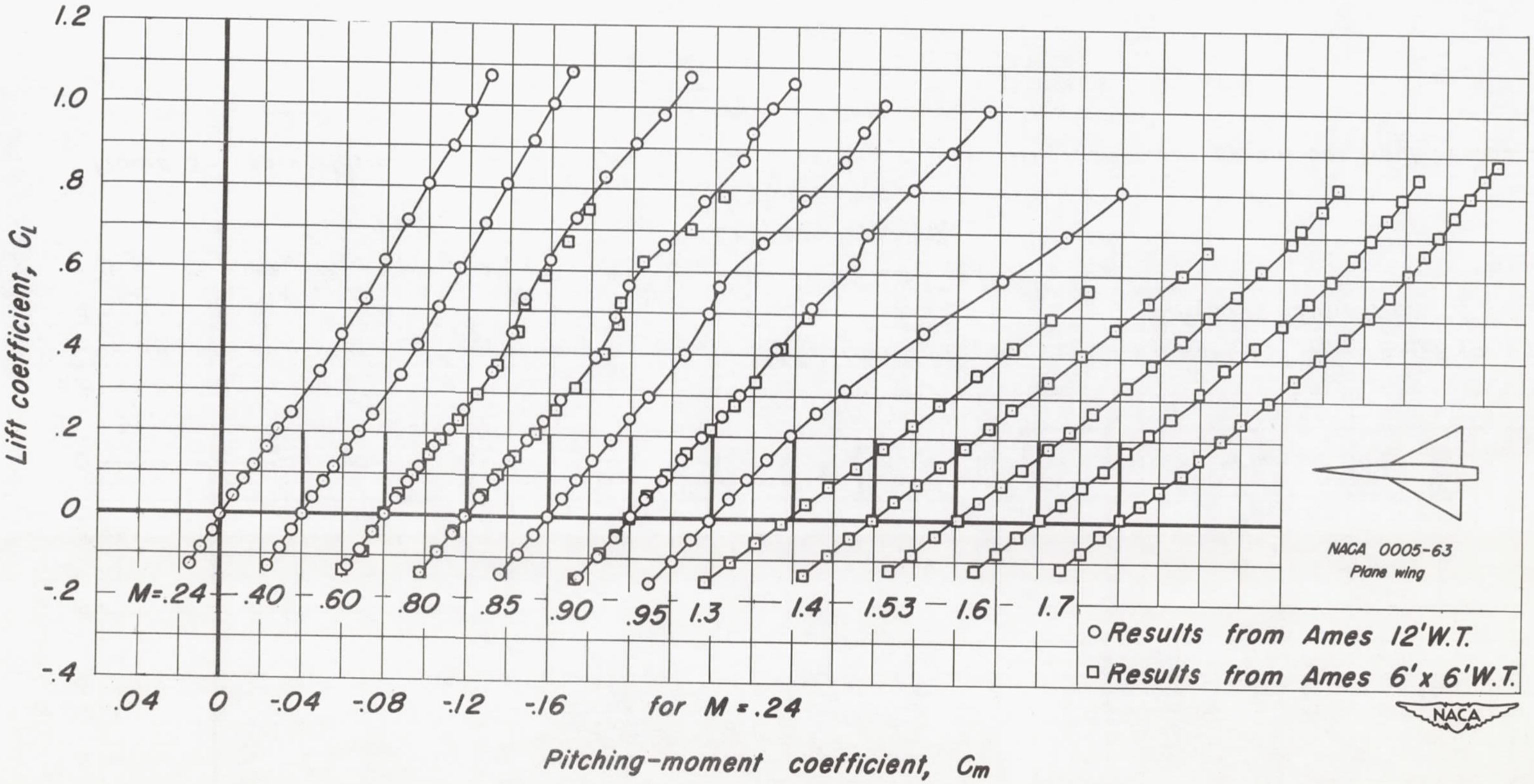


Figure 2.- Plan and front views of the model.



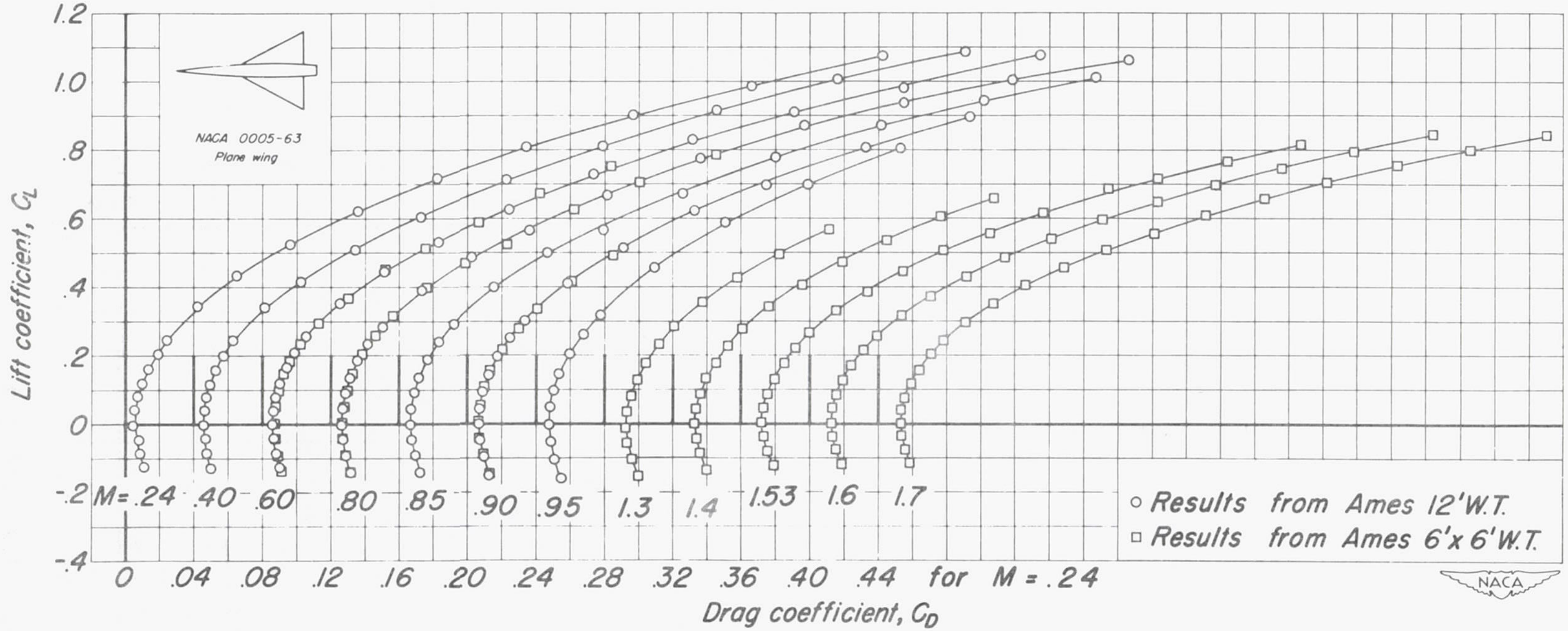
(a) C_L vs α.

Figure 3.- The variation of the aerodynamic characteristics with lift coefficient at various Mach numbers. Reynolds number, 3.0 million.



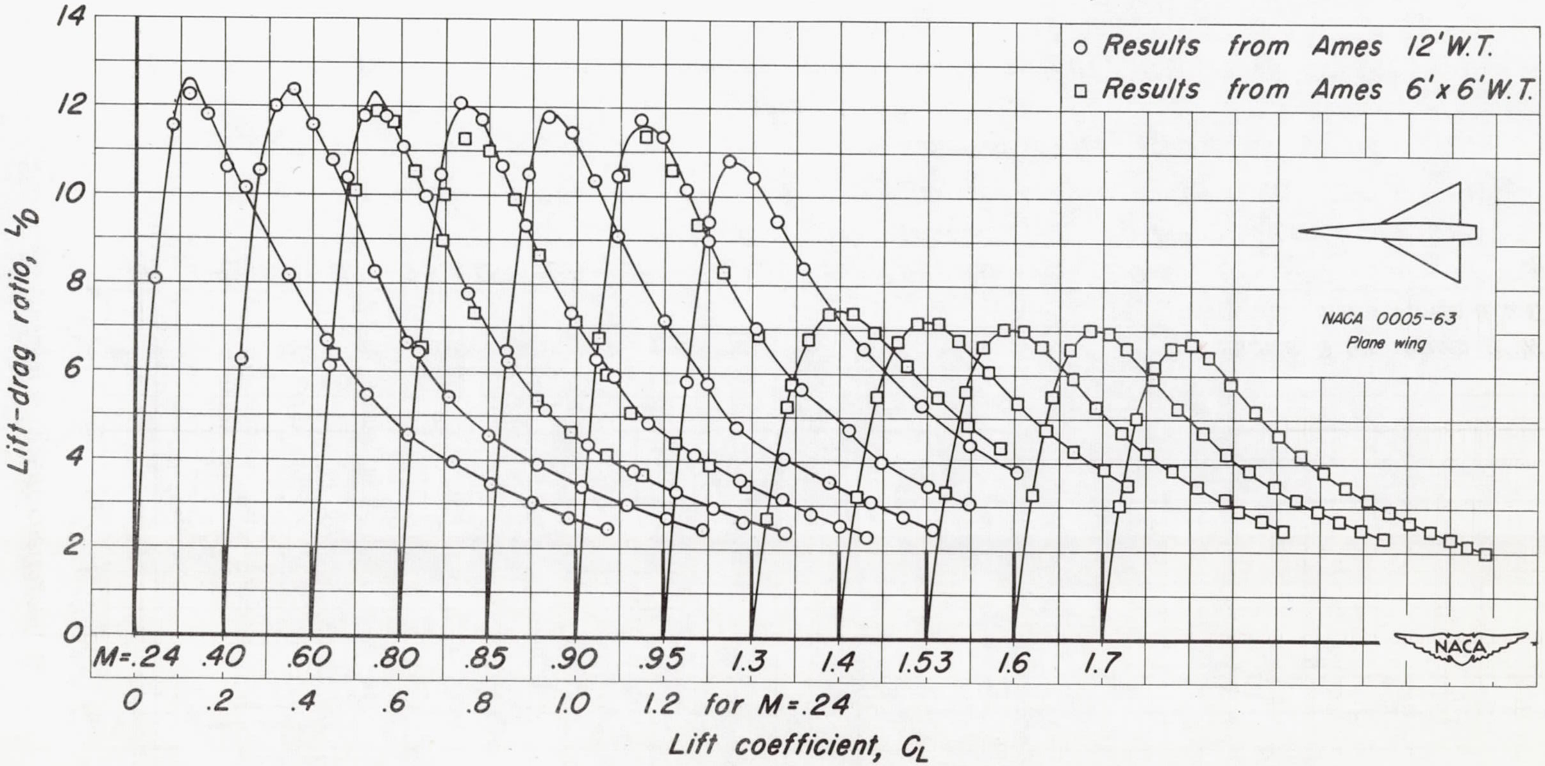
(b) C_L vs C_m.

Figure 3.- Continued.



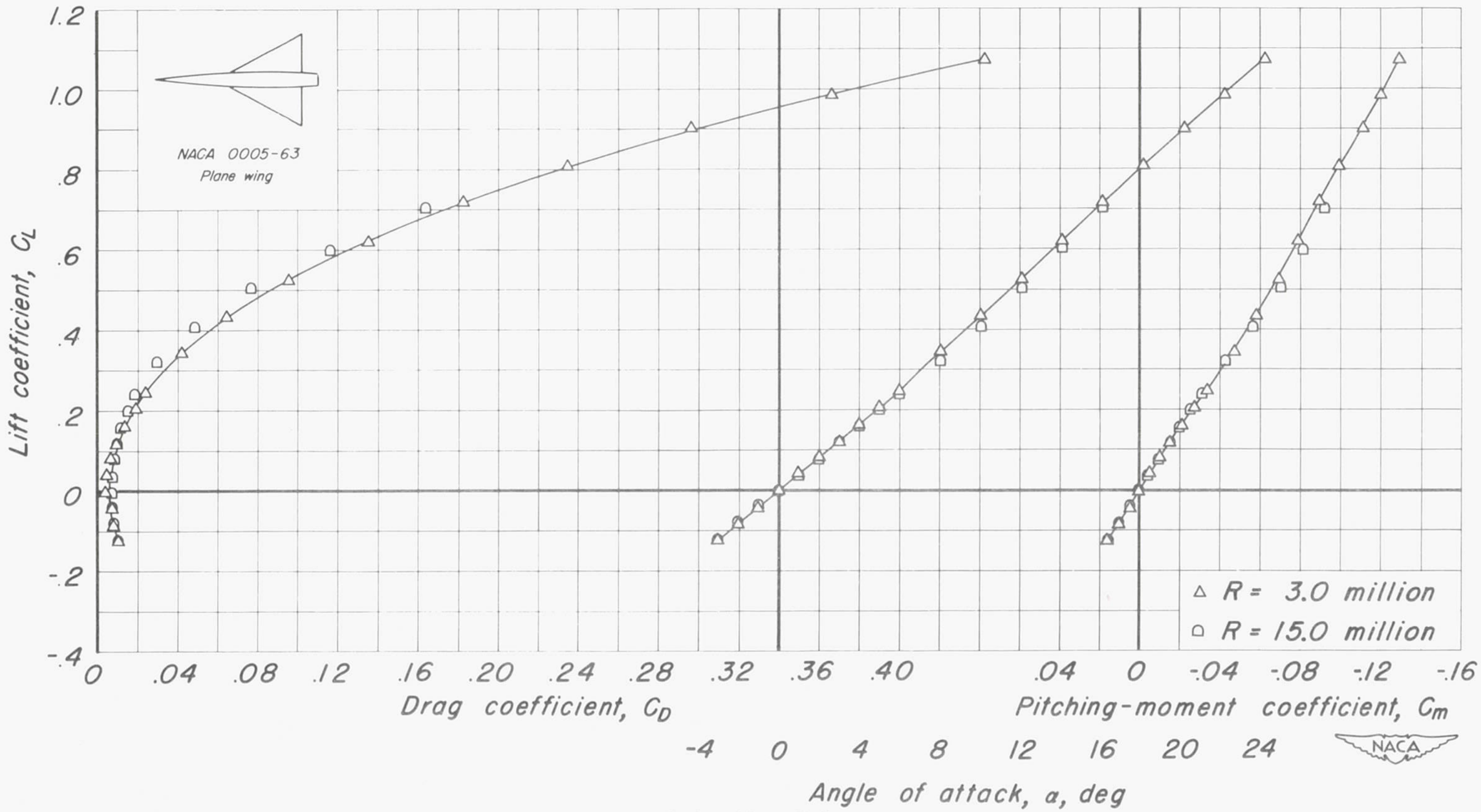
(c) C_L vs C_D.

Figure 3.- Continued.



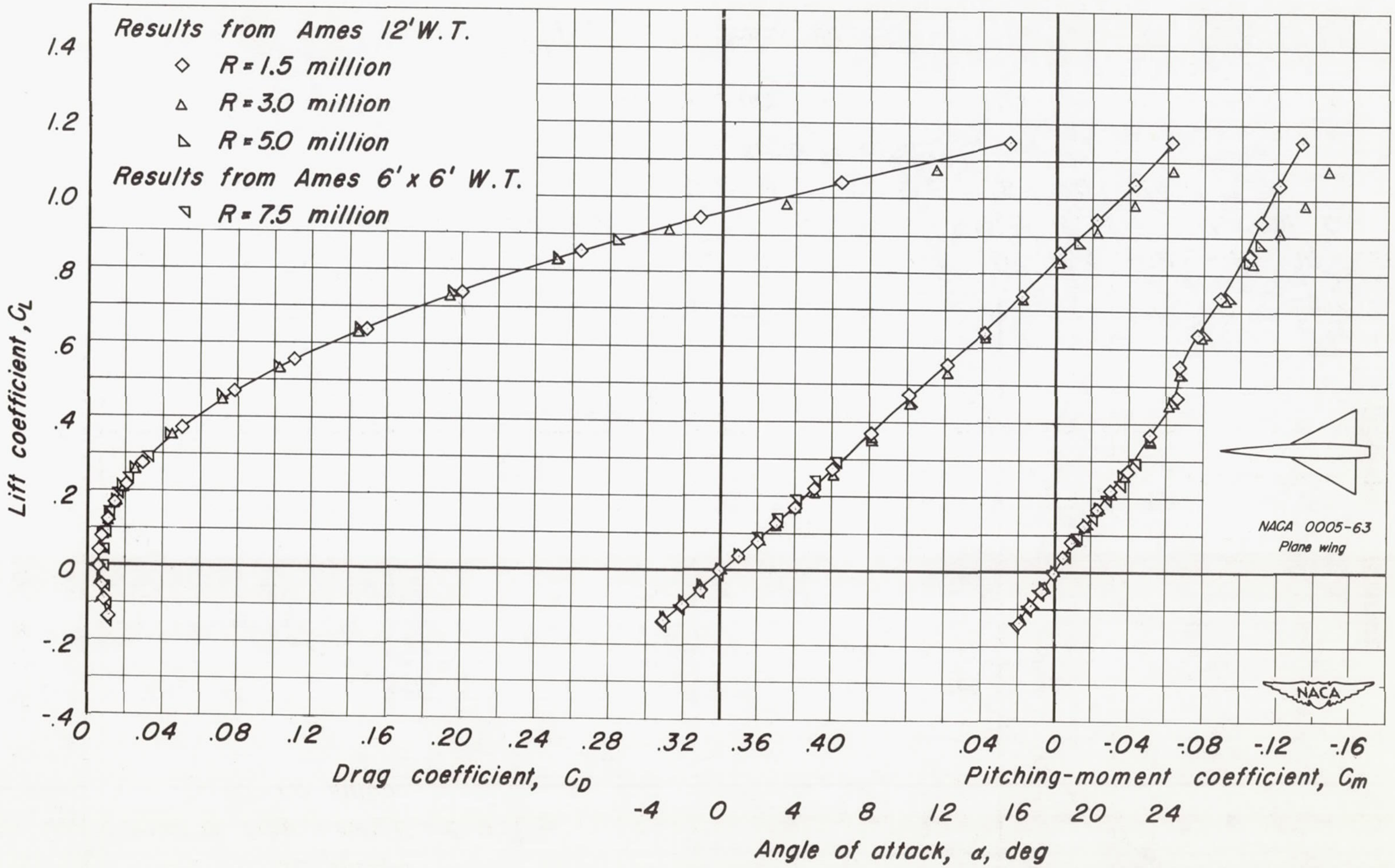
(d) L/D vs C_L.

Figure 3.- Concluded.



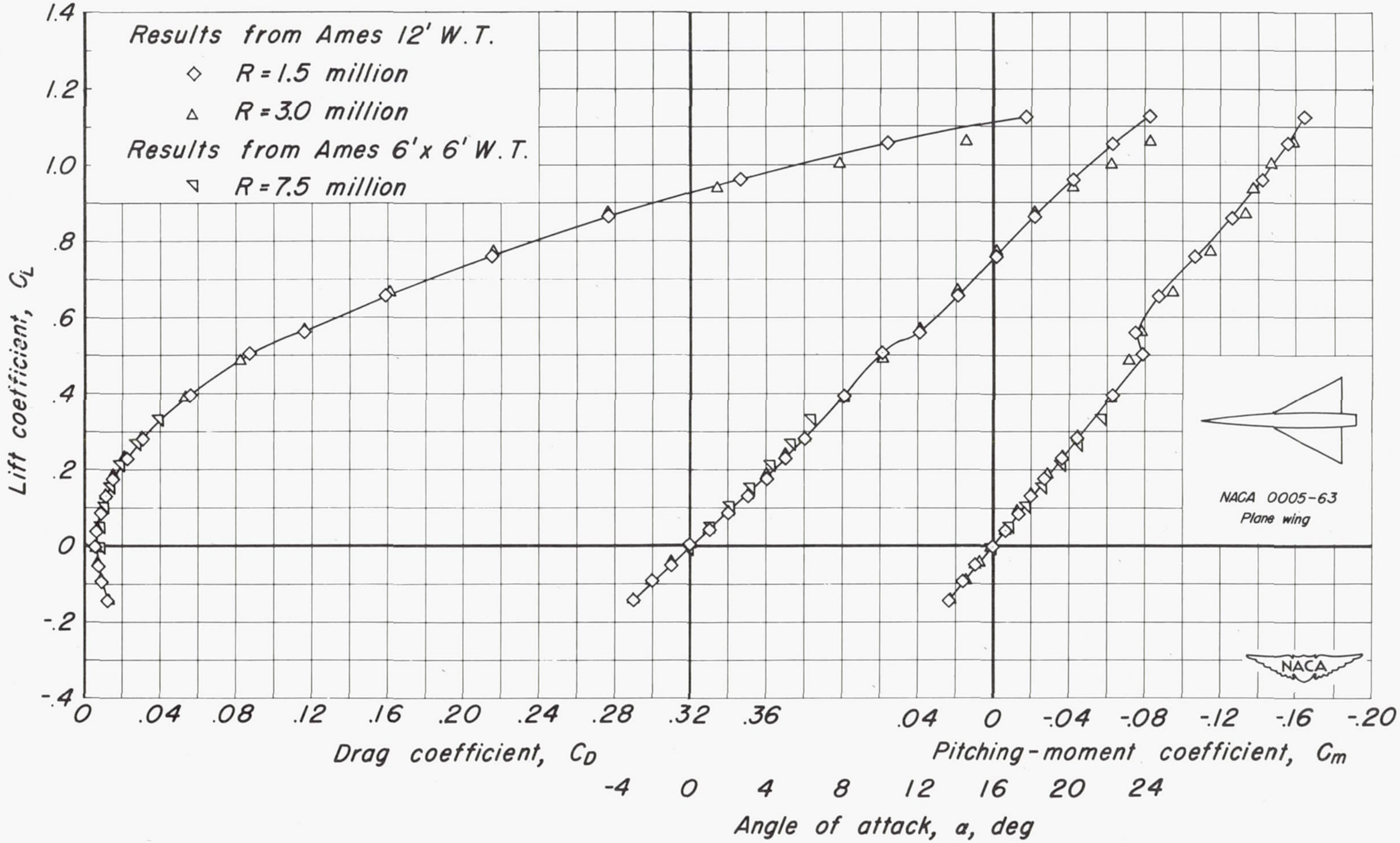
(a) M = .24.

Figure 4.- The variation of the aerodynamic characteristics with lift coefficient at various Reynolds numbers.



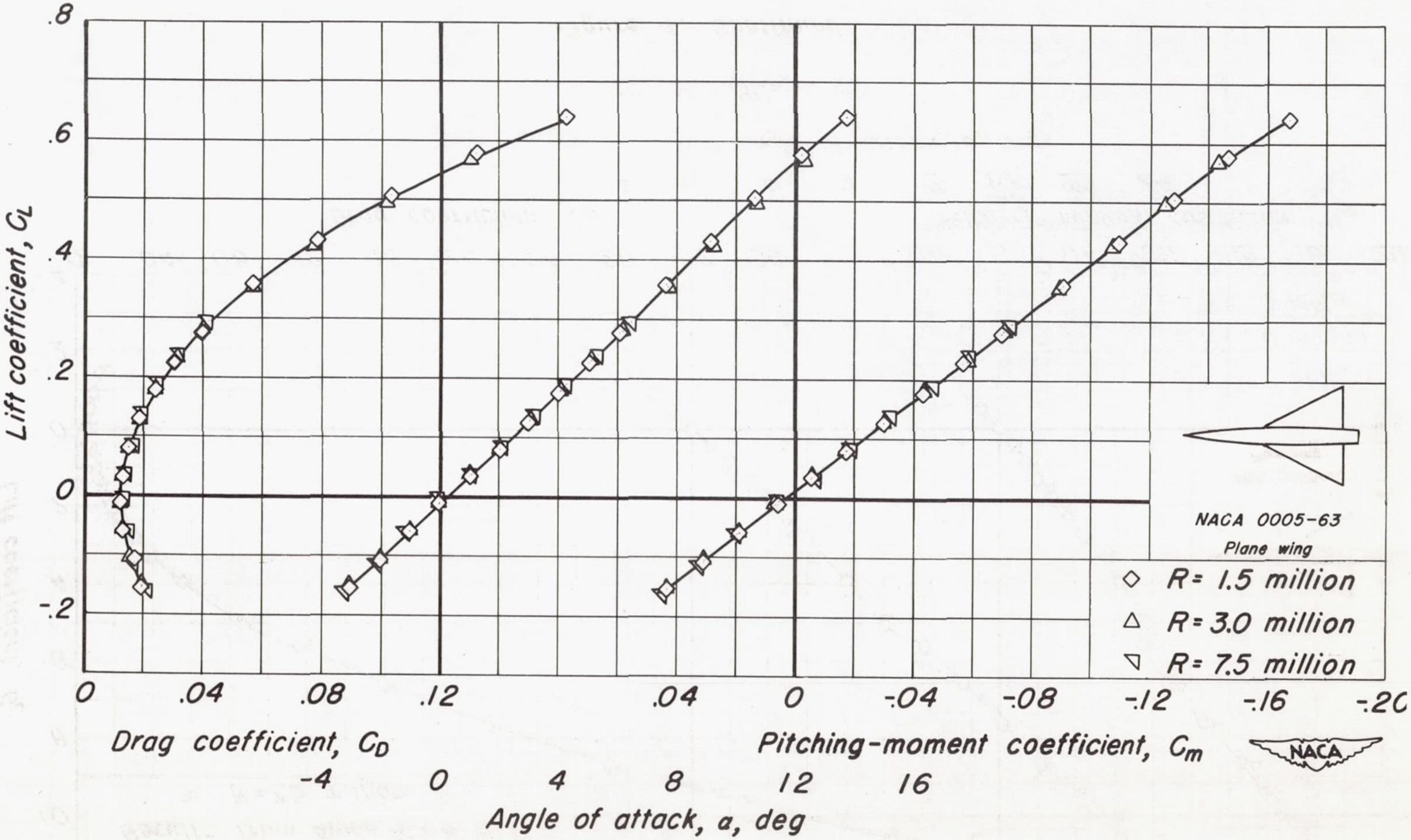
(b) M = .60.

Figure 4.- Continued.



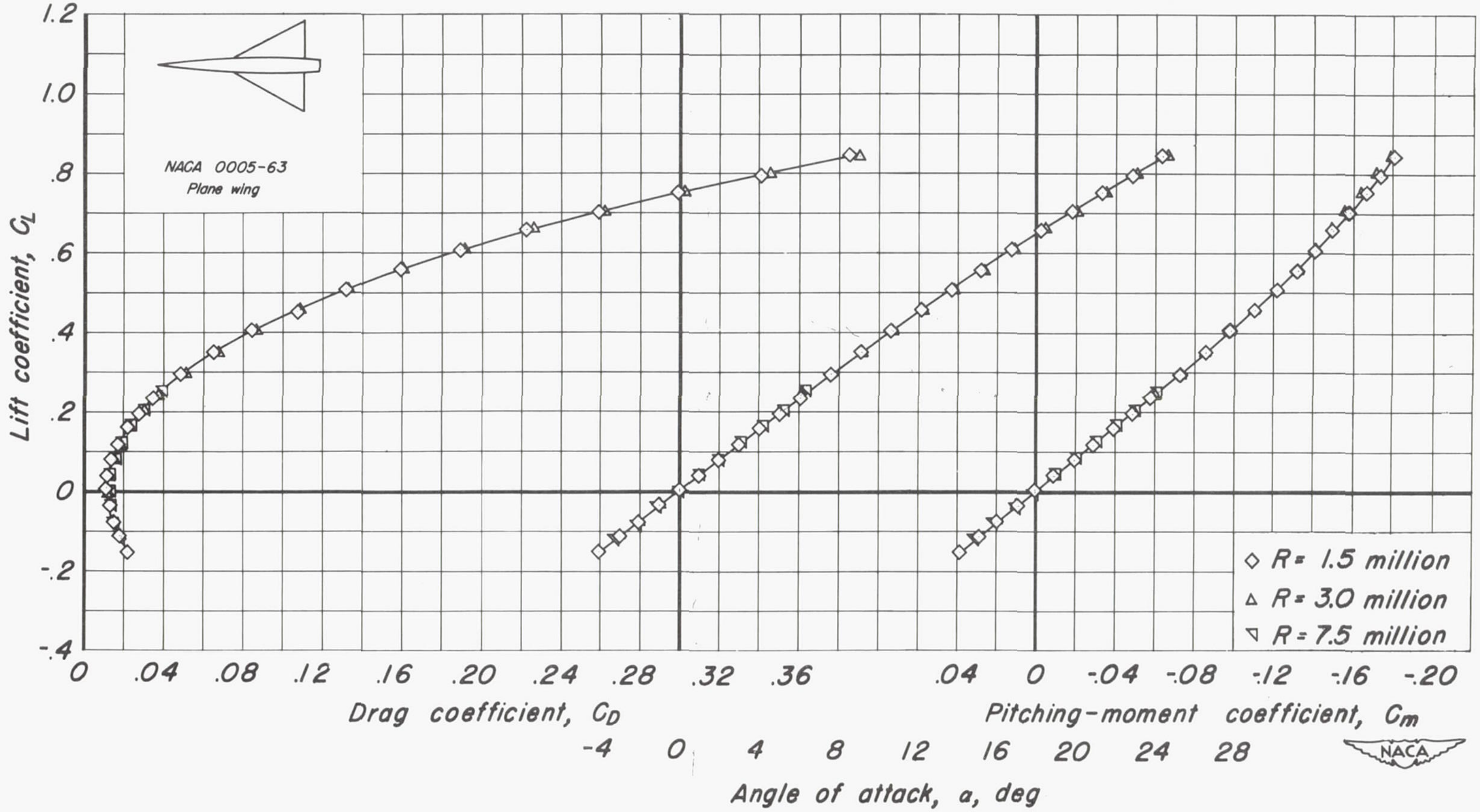
(c) M = .80.

Figure 4.- Continued.



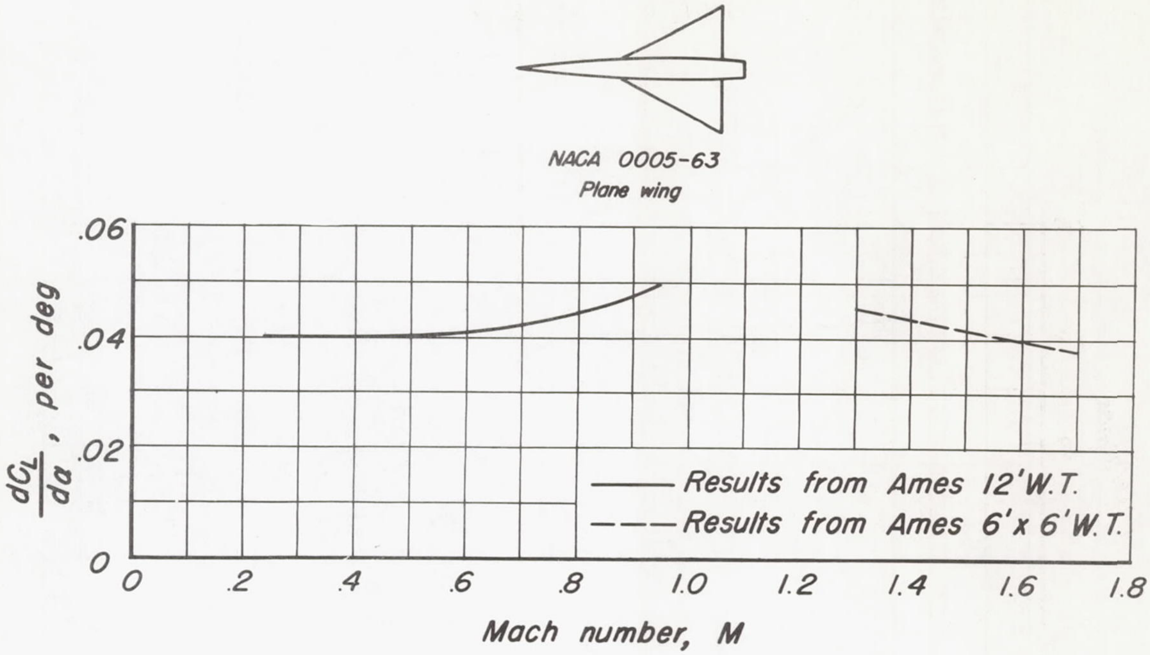
(d) M = 1.30.

Figure 4.- Continued.

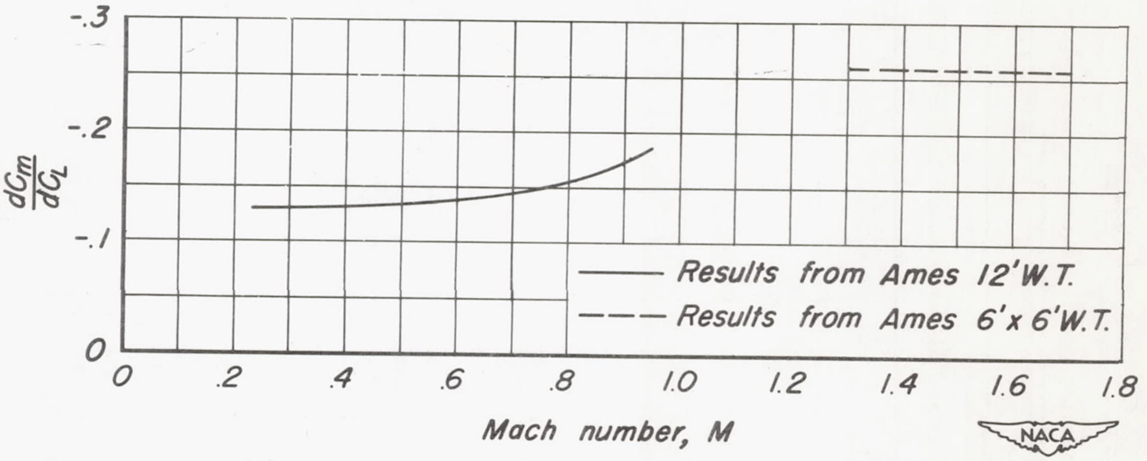


(e) M = 1.70.

Figure 4.– Concluded.

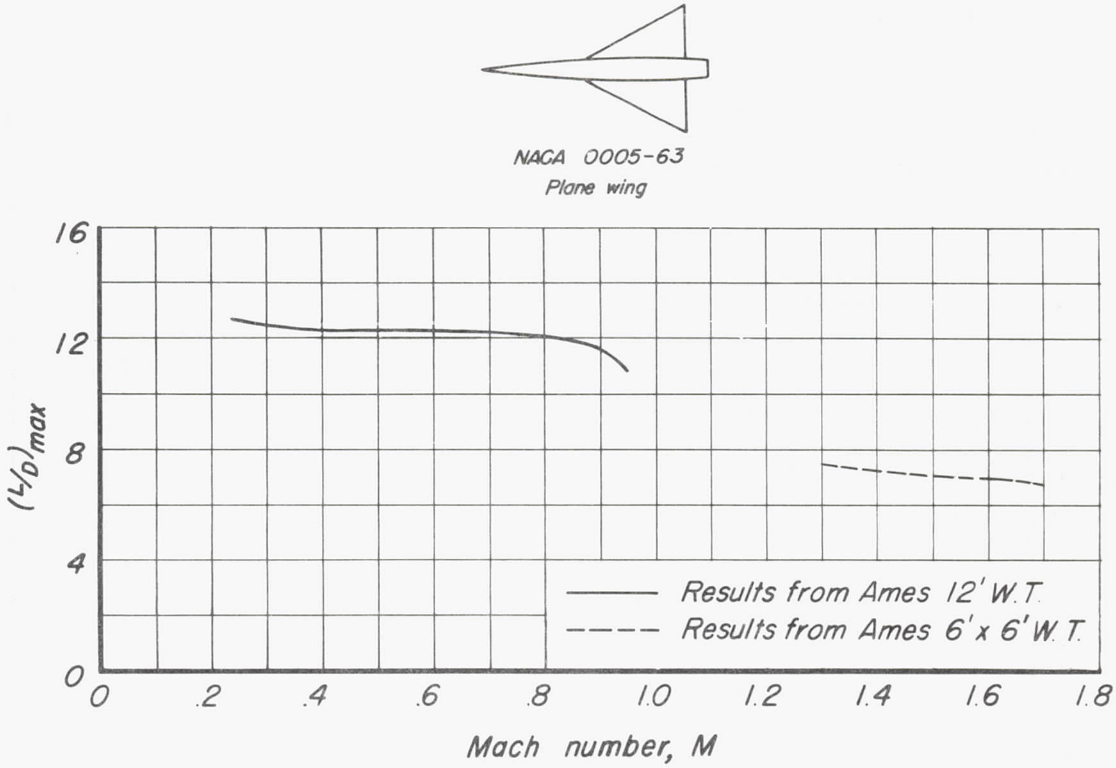


(a) $\frac{dC_L}{d\alpha}$ vs M.

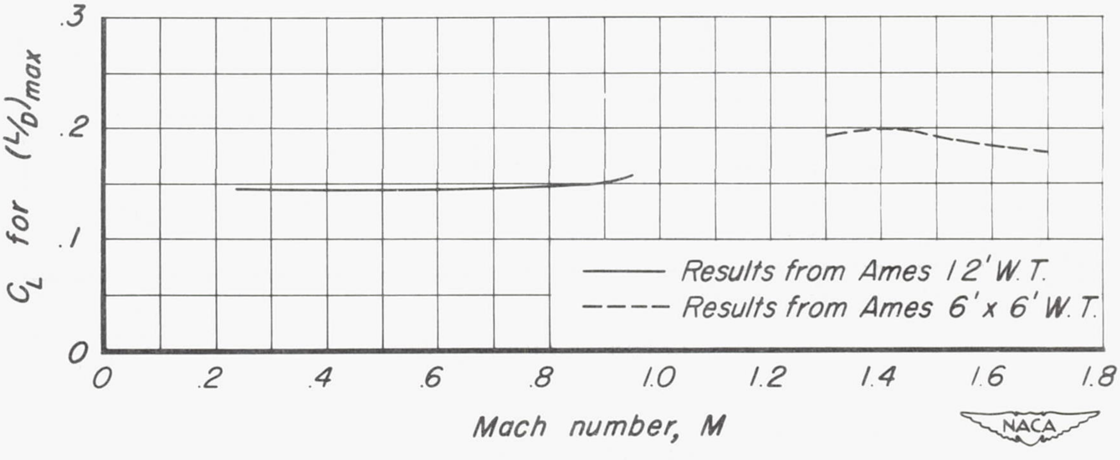


(b) $\frac{dC_m}{dC_L}$ vs M.

Figure 5.- Summary of aerodynamic characteristics as a function of Mach number. Reynolds number, 3.0 million.

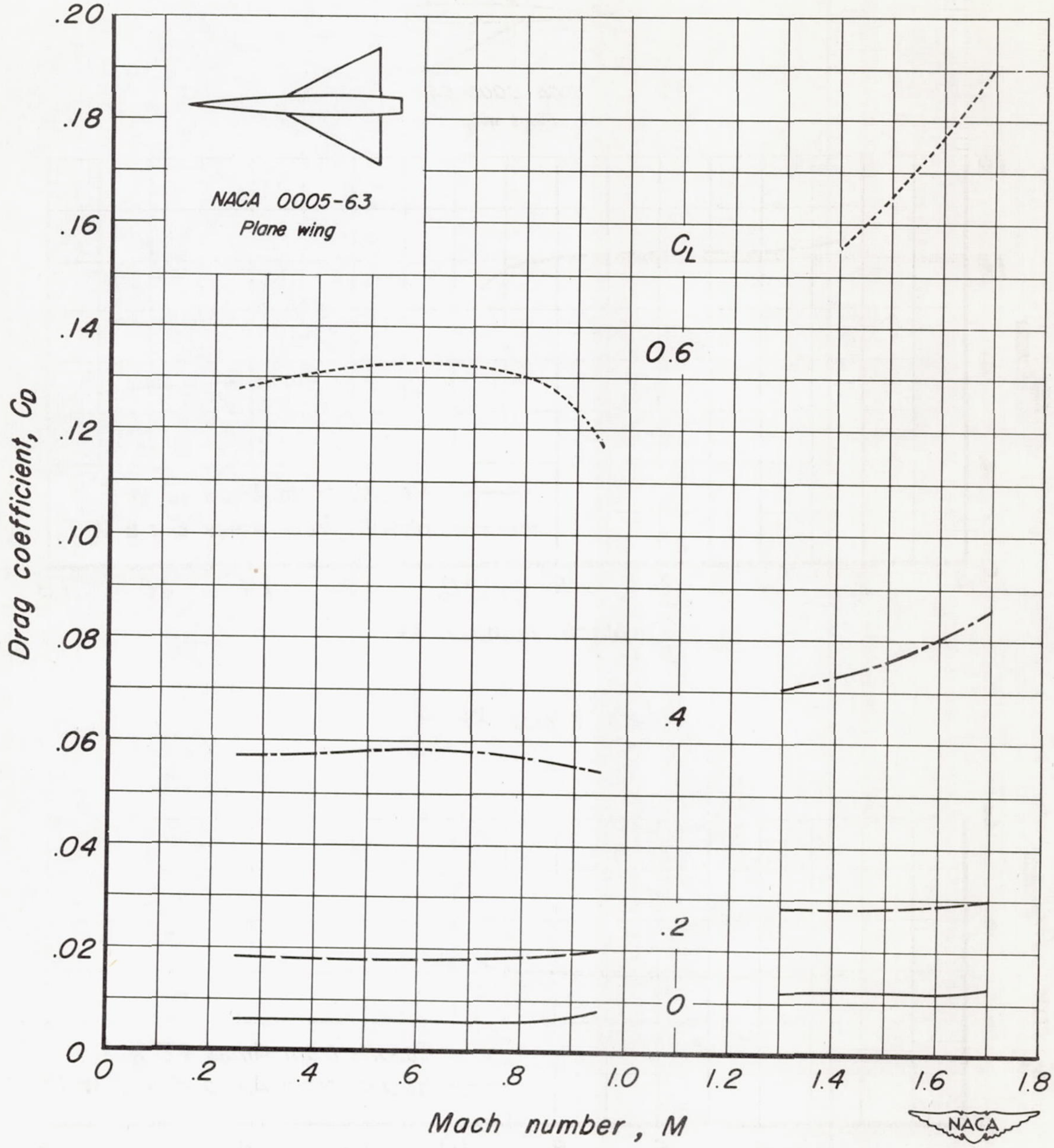


(c) $(L/D)_{max}$ vs M.



(d) C_L for $(L/D)_{max}$ vs M.

Figure 5.- Continued.



(e) C_D vs M.

Figure 5.– Concluded.